REPORT No. 668

WIND-TUNNEL INVESTIGATION OF N. A. C. A. 23012, 23021, AND 23030 AIRFOILS WITH VARIOUS SIZES OF SPLIT FLAP

By Carl J. Wenzinger and Thomas A. Harris

SUMMARY

An investigation has been made in the N. A. C. A. 7- by 10-foot wind tunnel of large-chord N. A. C. A. 23012, 23021, and 23030 airfoils with split flaps 10, 20, 30, and 40 percent of the wing chord to determine the section aerodynamic characteristics of the airfoils as affected by airfoil thickness, flap chord, and flap deflection. The complete section aerodynamic characteristics of all the combinations tested are given in the form of graphs of lift, drag, and pitching-moment coefficients, and certain applications to aerodynamic design are discussed.

The final maximum lift coefficients for the three airfoils tested with the $0.20c_w$ flap were about equal. For the airfoils with the $0.10c_w$ flap, the maximum lift coefficient decreased with airfoil thickness; for the airfoils with the $0.30c_w$ or $0.40c_w$ flaps, the maximum lift coefficient increased with airfoil thickness to a maximum value of 2.94. Within the range covered, the increment of maximum lift coefficient due to the split flaps was practically independent of Reynolds Number. The increase in minimum profile-drag coefficient with airfoil thickness was large, being about twice as great for the N. A. C. A. 23030 as for the 23012 plain airfoil.

INTRODUCTION

The National Advisory Committee for Aeronautics is undertaking an extensive investigation of various high-lift arrangements to furnish information applicable to the design of wing combinations for the improvement of the safety and the performance of airplanes. Thus far, most of the tests have been made with wings having a thickness 12 percent of the wing chord and having the Clark Y or the N. A. C. A. 23012 profile. It appears very desirable at the present time, however, to extend the investigation to include wings having other thicknesses and also other airfoil profiles. The present report describes the results obtained from tests in the 7- by 10-foot wind tunnel of airfoils of various thicknesses equipped with high-lift devices.

The investigation was made of airfoils having thicknesses from 12 to 30 percent of the wing chord; these thicknesses are believed to cover the range likely to be met with in practice. Airfoil sections of the N. A. C. A. 230 series were used because they appear to be generally satisfactory for most purposes. The high-lift device investigated with the airfoils of various thicknesses was the simple split flap, which is used as a basis of comparison with other high-lift devices. Flaps ranging in chord from 10 to 40 percent of the wing chord were tested on each airfoil. These tests are expected to be followed at a later date with tests of slotted flaps on similar airfoils.

MODELS

PLAIN AIRFOILS

Three basic wings, or plain airfoils, were used in these tests; each had a chord of 3 feet and a span of 7 feet. The models were constructed of laminated wood and were built to the N. A. C. A. 23012, 23021, and 23030 profiles. The thickness of each of these airfoils is, respectively, 12, 21, and 30 percent of the wing chord, c_w. The ordinates for each of the three airfoils are listed in table I. The N. A. C. A. 23012 airfoil, which had been previously used for the investigation described in reference 1, was already available.

FLAPS

Four simple split flaps extending along the entire span were used with each model. The flap chords, c_f, were $0.10c_w$, $0.20c_w$, $0.30c_w$, and $0.40c_w$ and were believed likely to cover the range of sizes that might be used in practice. (See figs. 1, 2, and 3.) The flaps were built of plywood braced at several points along the span and were arranged for setting at deflections from 0° to 105° down. The flap deflection, δ_f, is measured between the lower surface of each airfoil and the flap, as shown in figures 1, 2, and 3.

TESTS

The models were mounted in the closed test section of the N. A. C. A. 7- by 10-foot wind tunnel so as to span the jet completely except for small clearances at each end. (See references 1 and 2.) The main airfoil was rigidly attached to the balance frame by torque tubes, which extended through the upper and the lower boundaries of the tunnel. The angle of attack of the model was set from outside the tunnel by rotating the torque tubes with a calibrated electric drive. Approximately two-dimensional flow is obtained with this type of installation and the section characteristics of the model under test can be determined.

A dynamic pressure of 16.37 pounds per square foot was maintained for most of the tests, which corresponds to a velocity of 80 miles per hour under standard atmospheric conditions and to an average test Reynolds Number of about 2,190,000. Because of the

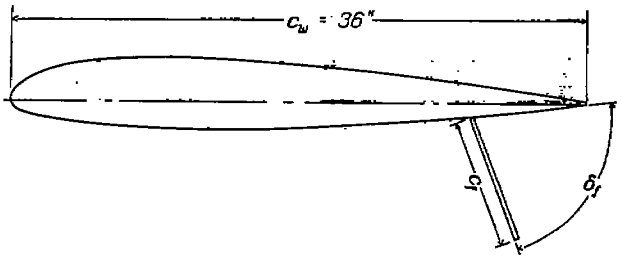


FIGURE 1.—Section of N. A. C. A. 23012 airfoil with split flaps. c_f=0.10c_w, 0.20c_w, 0.30c_w, and 0.40c_w.

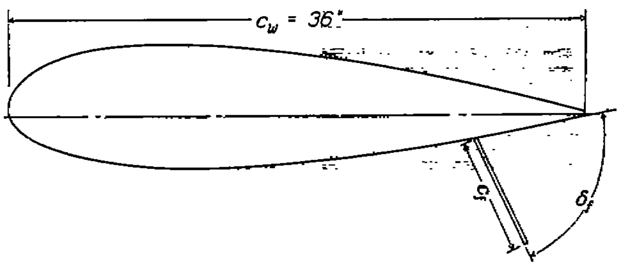


FIGURE 2.—Section of N. A. C. A. 23021 airfoil with split flaps. c_f=0.10c_w, 0.20c_w, 0.30c_w, and 0.40c_w.

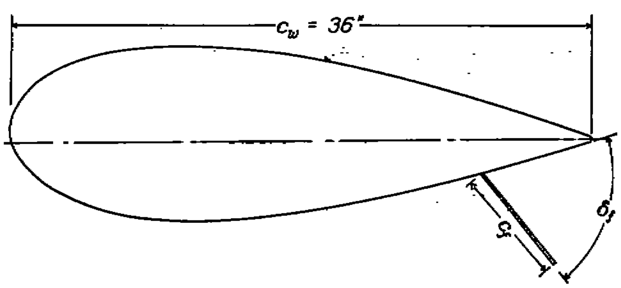


FIGURE 3.—Section of N. A. C. A. 23030 airfoil with split flaps. c_f=0.10c_w, 0.20c_w, 0.30c_w, and 0.40c_w.

turbulence in the wind tunnel, the effective Reynolds Number, R_e, was approximately 3,500,000. For all tests, R_e is based on the chord of the airfoil with the flap retracted and on a turbulence factor of 1.6 for the tunnel.

Each airfoil was tested by itself without the flap so that the characteristics of the plain airfoils could be determined. Each of the four split flaps was then tested on each of the three airfoils and deflected in 10° or 15° increments up to the deflection giving the highest value of the maximum lift coefficient.

An angle-of-attack range from −6° to the angle of attack for maximum lift was covered in 2° increments for each test. Lift, drag, and pitching moment were measured at each angle of attack.

RESULTS AND DISCUSSION

COEFFICIENTS

All test results are given in standard section nondimensional coefficient form for the airfoil and flap combinations corrected as explained in reference 1.

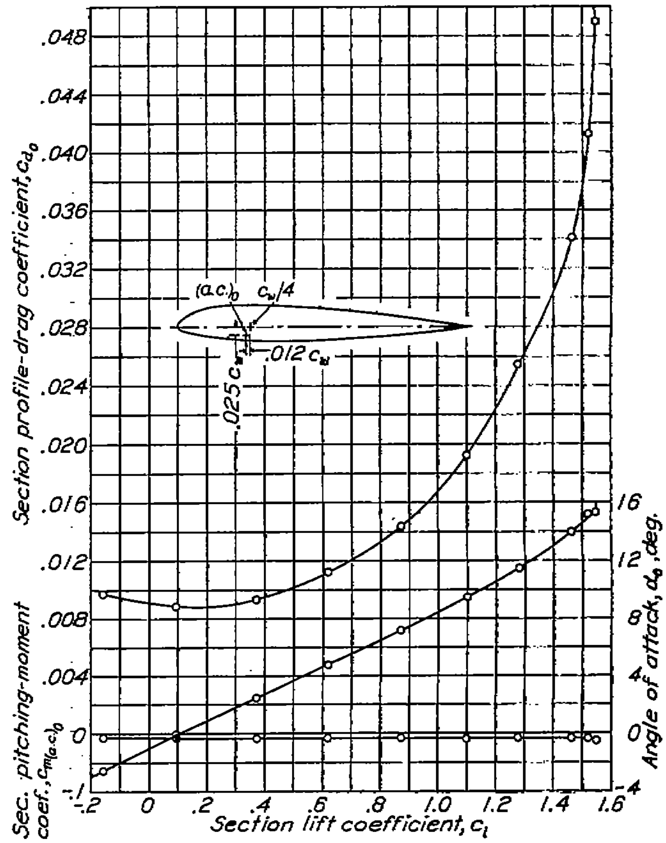


FIGURE 4.—Section aerodynamic characteristics of N. A. C. A. 23012 plain airfoil.

c_l, section lift coefficient, l/qc_w.

c_{d_0}, section profile-drag coefficient, d_0/qc_w.

$c_{m_{(a.c.)_0}}$, section pitching-moment coefficient about aerodynamic center of plain airfoil, $m_{(a.c.)_0}/qc_w^2$.

where

l is section lift.

d_0, section profile drag.

$m_{(a.c.)_0}$, section pitching moment.

q, dynamic pressure, $1/2\ \rho V^2$.

c_w, chord of basic airfoil with flap fully retracted.

and

α_0 is angle of attack for infinite aspect ratio.

δ_f, flap deflection.

PRECISION

The accuracy of the various measurements in the tests is believed to be within the following limits:

α_0	$\pm 0.1°$	$c_{d_{0(c_l=1.0)}}$	± 0.0006
$c_{l_{max}}$	± 0.03	$c_{d_{0(c_l=2.5)}}$	± 0.002
$c_{m_{(a.c.)_0}}$	± 0.003	δ_f	$\pm 0.2°$
$c_{d_{0_{min}}}$	± 0.0003		

SECTION AERODYNAMIC CHARACTERISTICS

Plain airfoils.—The section aerodynamic characteristics of the N. A. C. A. 23012 plain airfoil, as determined with the two-dimensional-flow installation, are shown

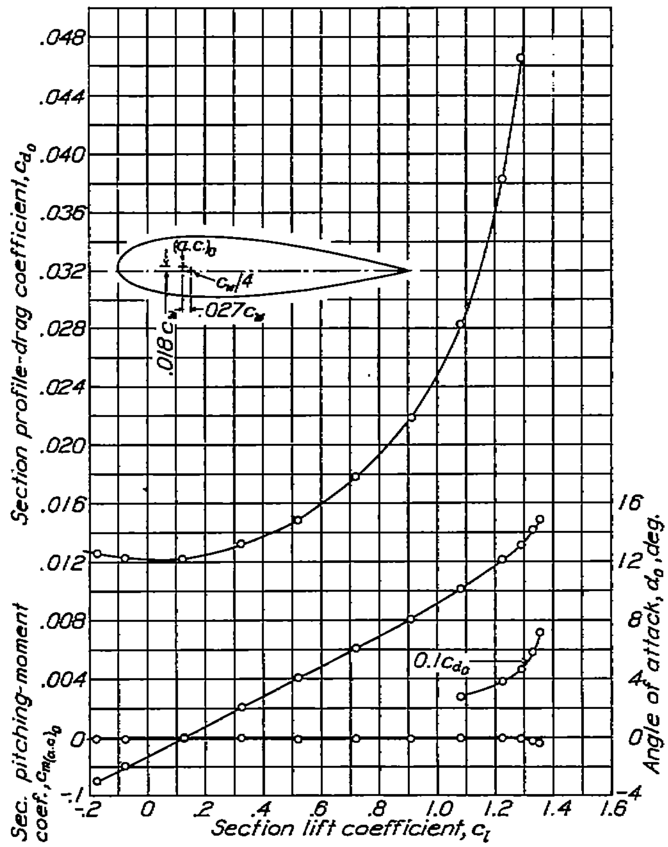


FIGURE 5.—Section aerodynamic characteristics of N. A. C. A. 23021 plain airfoil.

in figure 4. Similar results for the N. A. C. A. 23021 and the N. A. C. A. 23030 plain airfoils are given in figures 5 and 6, respectively. The data for the N. A. C. A. 23012 and 23021 airfoils are discussed in references 1 and 3, respectively, and therefore require no further discussion. The data for the N. A. C. A. 23030 airfoil, however, depart from the results of the thinner sections in several respects. The slope of the lift curve is only 0.068 as compared with about 0.105 for the N. A. C. A. 23012, although there is a marked increase in slope at angles of attack above 2°. The angle of attack for zero lift, however, is the same as for the N. A. C. A. 23012 and 23021 airfoils. The relatively flat-top lift curve given by the N. A. C. A. 23030 airfoil is probably typical of very thick airfoils. Its pitching-moment

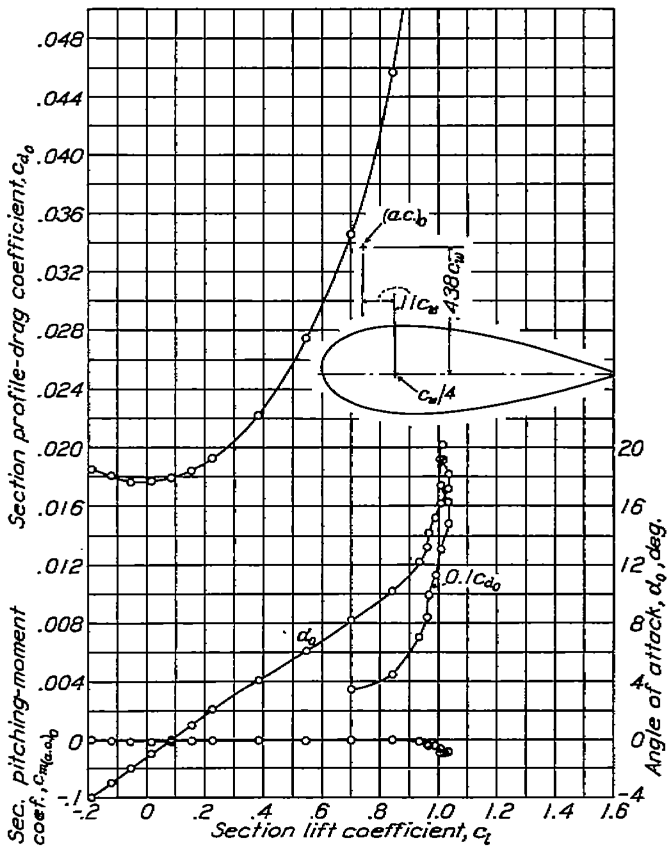


FIGURE 6.—Section aerodynamic characteristics of N. A. C. A. 23030 plain airfoil.

coefficient about the aerodynamic center is −0.002 compared with −0.003 for the N. A. C. A. 23021 and −0.009 for the N. A. C. A. 23012. The most marked change is the position of the aerodynamic center of the plain airfoil; it is 11 percent of the chord ahead of the quarter-chord point of the wing and about 44 percent of the chord above the chord line.

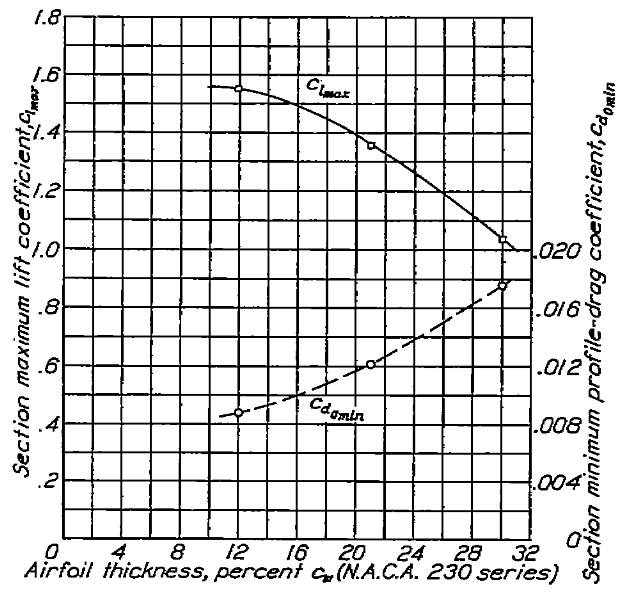
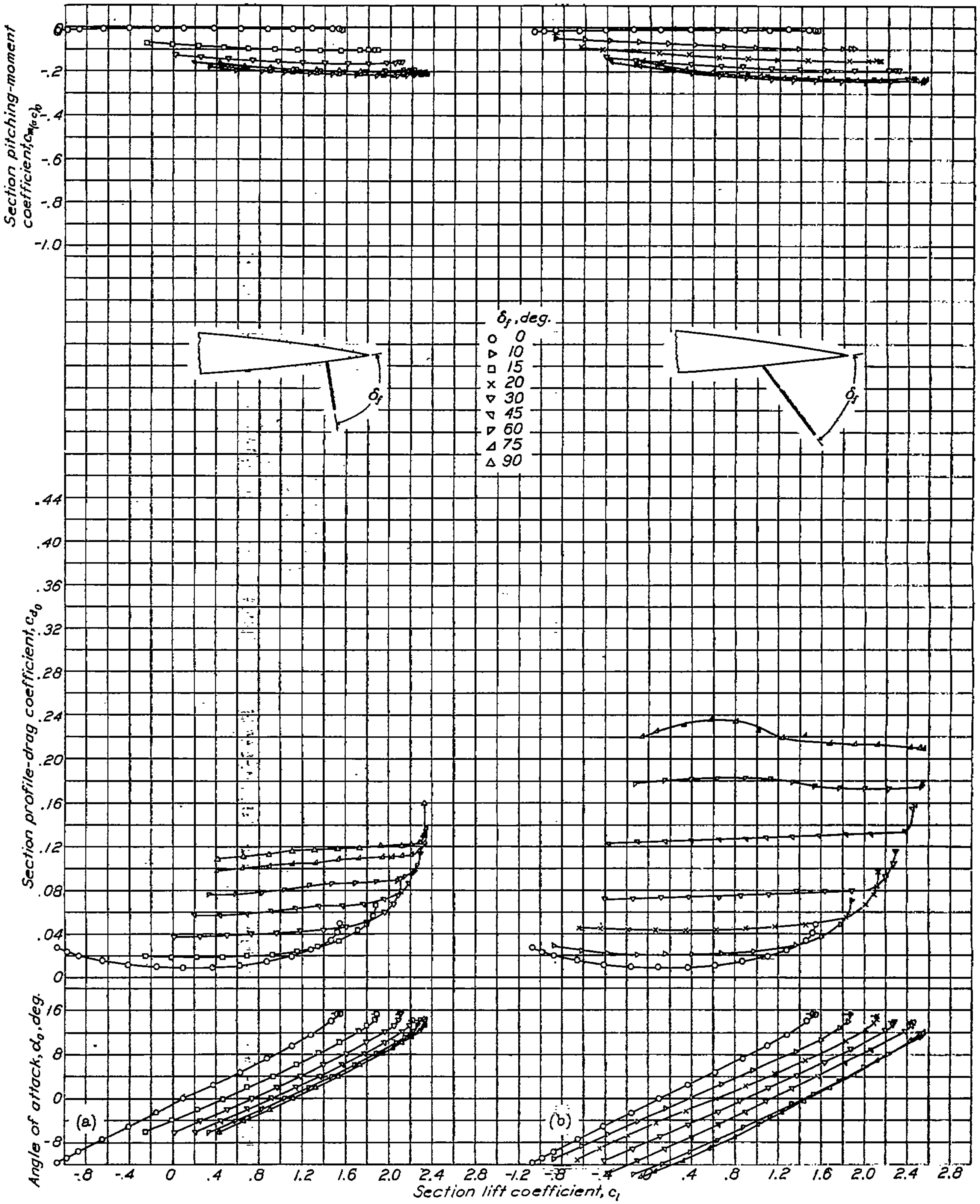


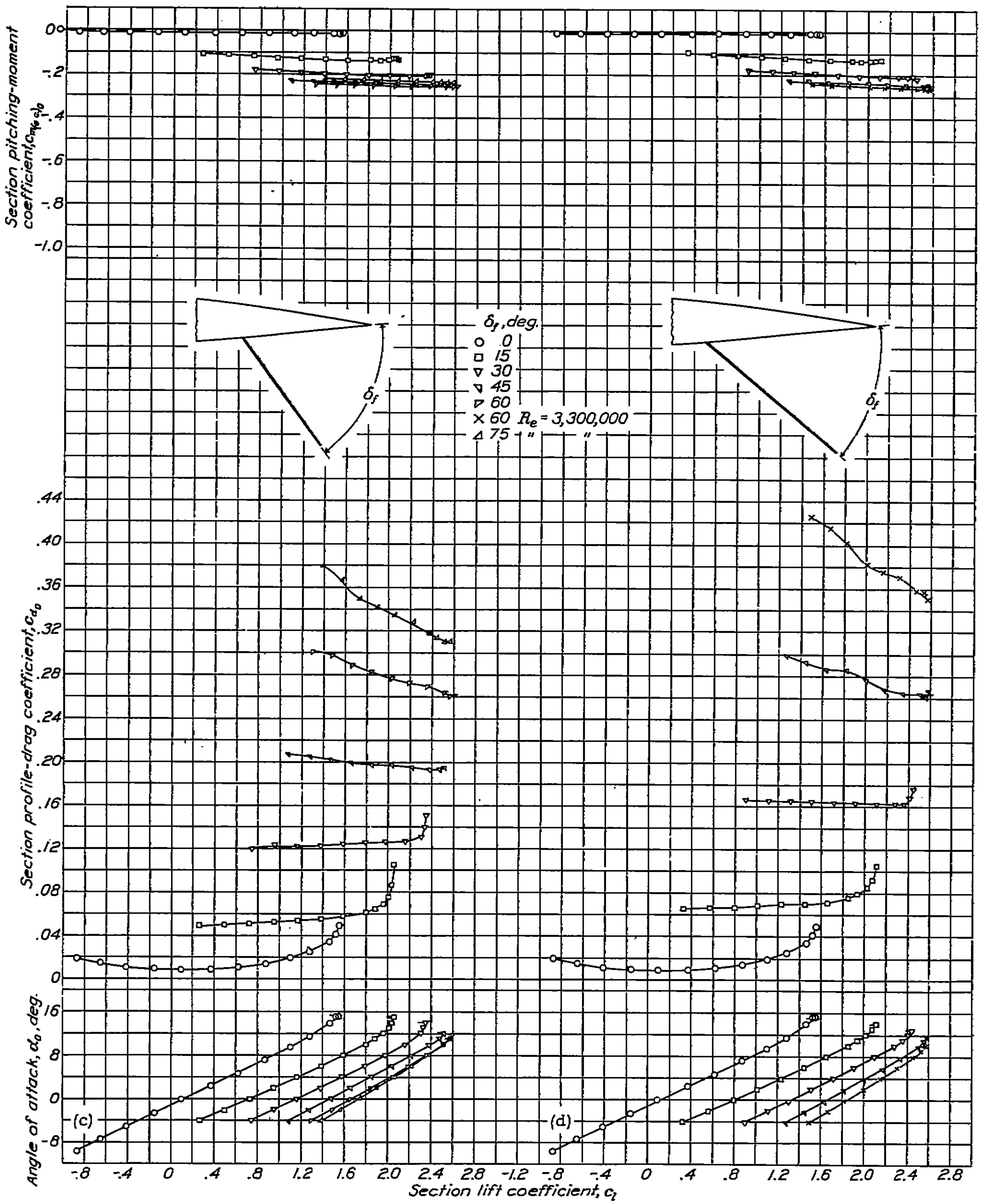
FIGURE 7.—Effect of thickness of plain airfoils on maximum lift and minimum drag.



(a) The $0.10c_w$ split flap.

(b) The $0.20c_w$ split flap.

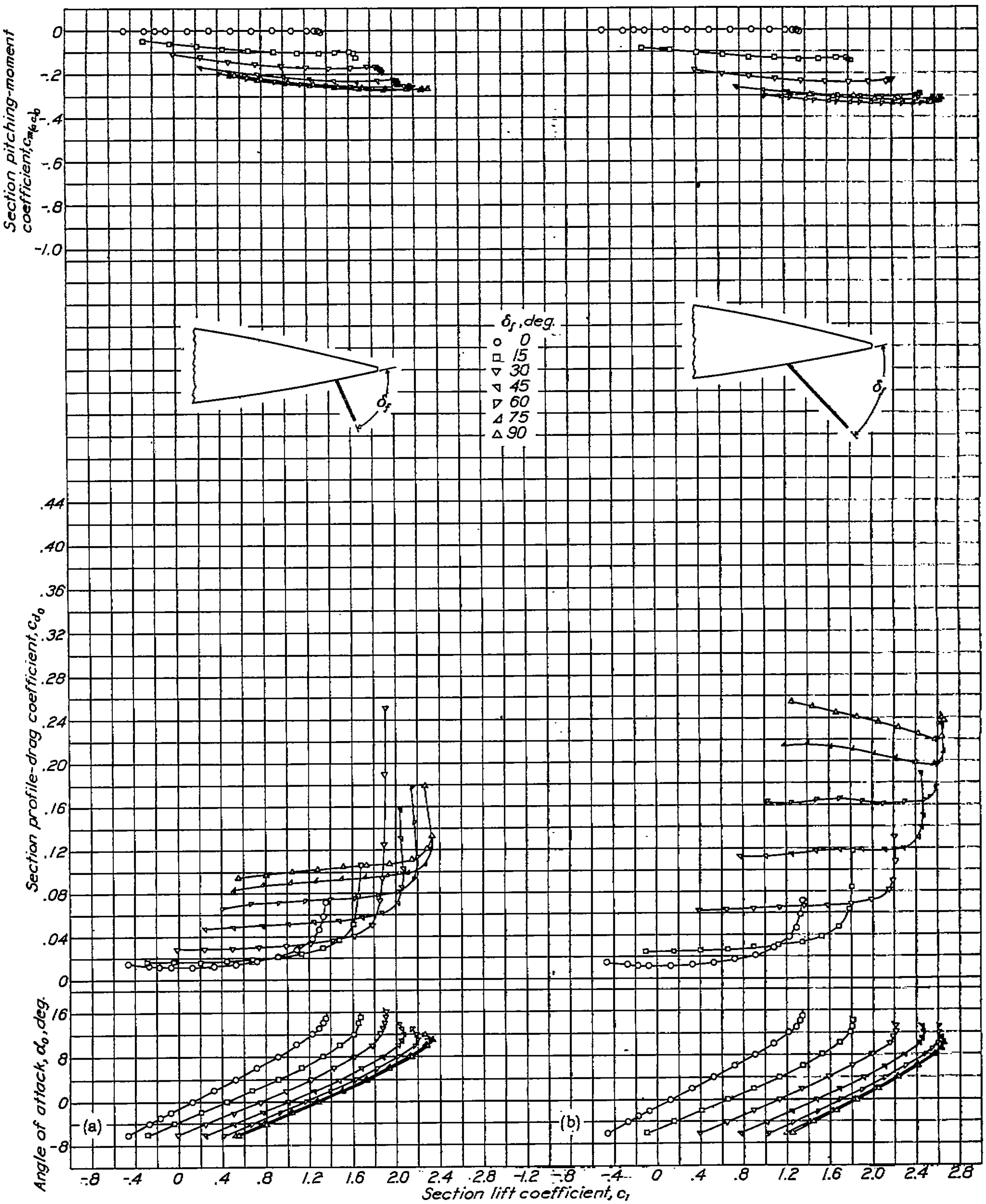
FIGURE 8.—Section aerodynamic characteristics of N. A. C. A. 23012 airfoil with various sizes of split flap.



(c) The $0.30c_w$ split flap.

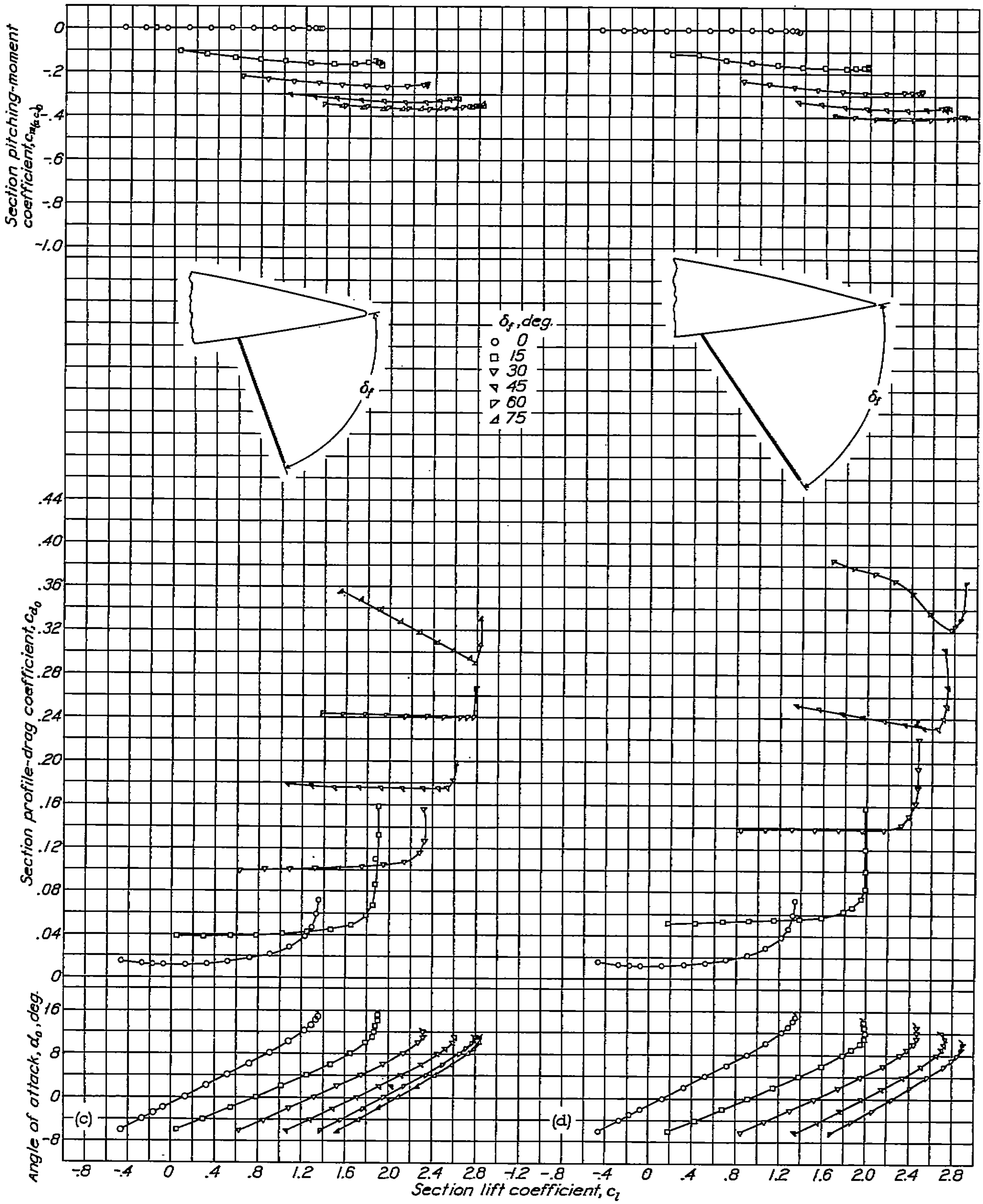
(d) The $0.40c_w$ split flap.

FIGURE 8.—Continued. Section aerodynamic characteristics of N. A. C. A. 23012 airfoil with various sizes of split flap.



(a) The $0.10c_w$ split flap. (b) The $0.20c_w$ split flap.

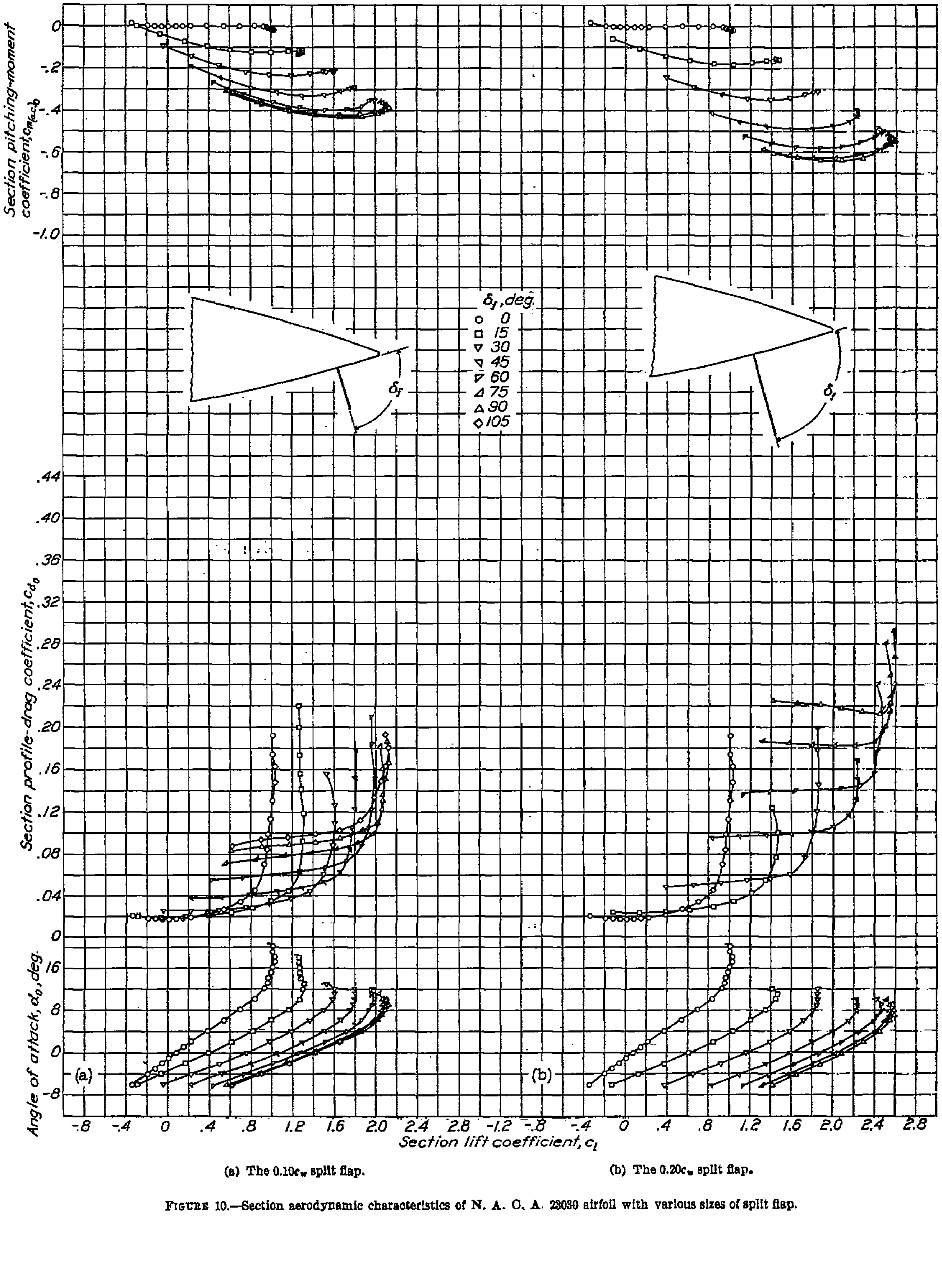
FIGURE 9.—Section aerodynamic characteristics of N. A. C. A. 23021 airfoil with various sizes of split flap.



(c) The 0.30c_w split flap.

(d) The 0.40c_w split flap.

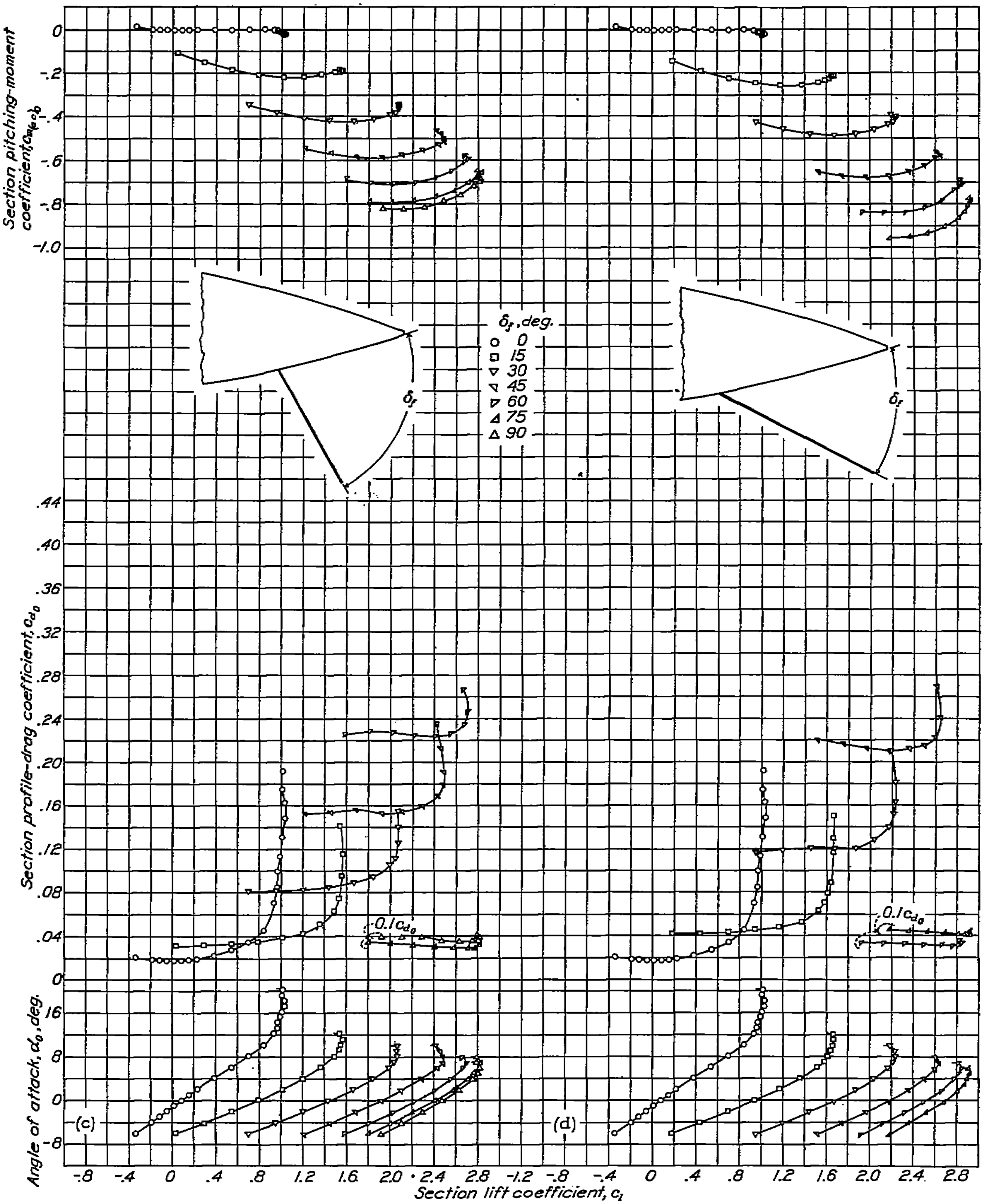
FIGURE 9.—Continued. Section aerodynamic characteristics of N. A. C. A. 23021 airfoil with various sizes of split flap.



(a) The $0.10c_w$ split flap.

(b) The $0.20c_w$ split flap.

FIGURE 10.—Section aerodynamic characteristics of N. A. C. A. 23030 airfoil with various sizes of split flap.



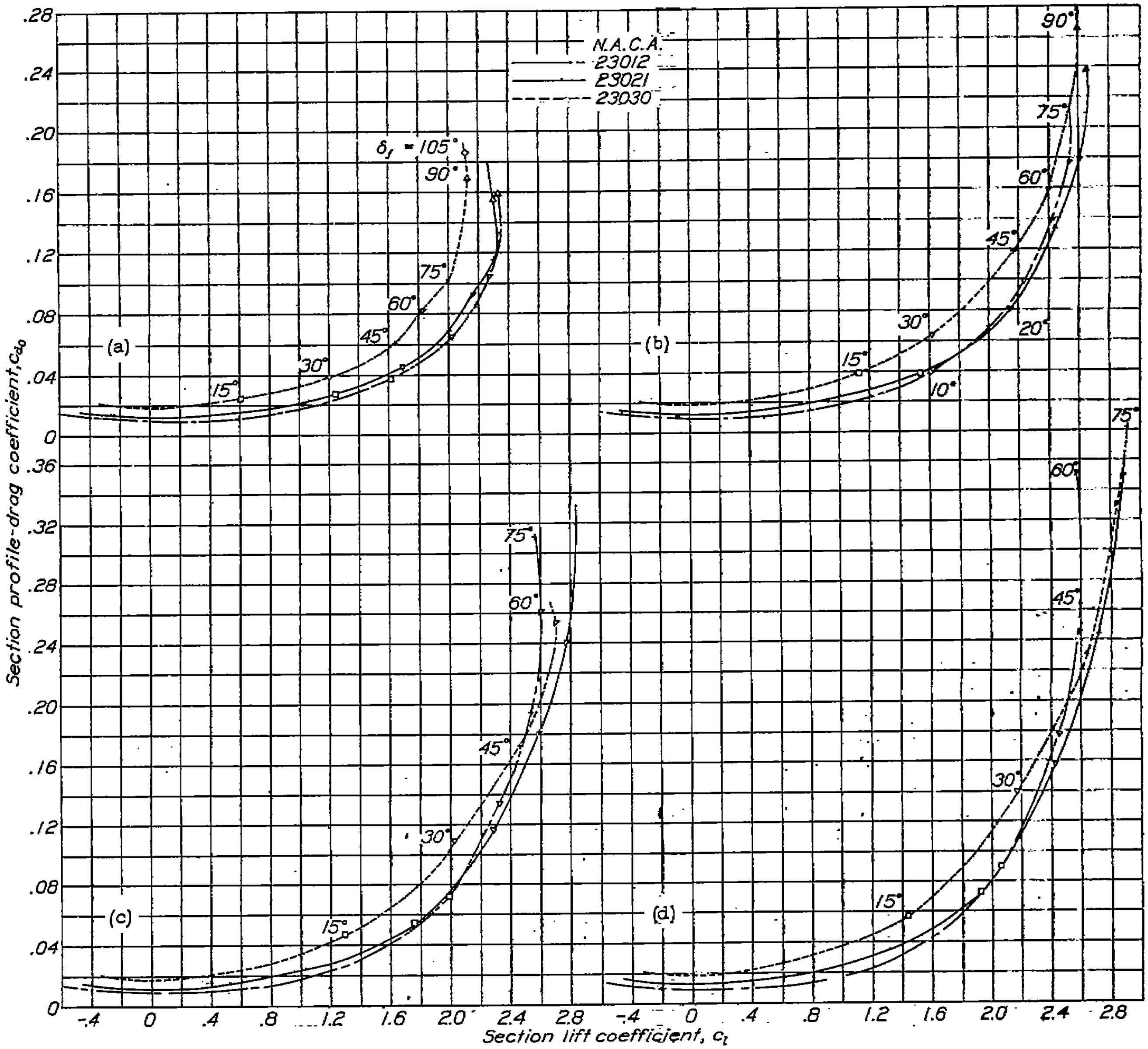
(c) The $0.30c_w$ split flap.

(d) The $0.40c_w$ split flap.

FIGURE 10.—Continued. Section aerodynamic characteristics of N. A. C. A. 23030 airfoil with various sizes of split flap.

The effects of a change in thickness of the plain airfoils on the minimum profile-drag coefficients and on the maximum lift coefficients are indicated in figure 7 for an effective Reynolds Number of 3,500,000. Although the minimum profile-drag coefficient increases rapidly with airfoil thickness and is nearly twice as great for the N. A. C. A. 23030 as for the N. A. C. A. 23012 airfoil (see fig. 7), it may be that structural considerations will more than overbalance this drag increase in application to a given design. In other words, the probability should not be overlooked of actually obtaining desired characteristics with the thick sections because of the possibility of housing parts of the airplane entirely within the wing, which would be impossible with the thinner sections.

Airfoils with flaps.—The section aerodynamic characteristics of the N. A. C. A. 23012 airfoil with the $0.10c_w$, the $0.20c_w$, the $0.30c_w$, and the $0.40c_w$ split flaps are shown in figure 8. All these data were obtained at an effective Reynolds Number of 3,500,000, except as noted on the figure. The lift curves have about the same slopes as they did for the plain airfoils. The angle of attack for maximum lift decreases from about 15° with the flap neutral to about 14° with the flap down 30°. With the flap down 60° or 75°, however, the angle of attack for maximum lift is only about 10° or 12°, a change of 5° or 3° from the plain airfoil. Changes of this magnitude in the angle of attack for maximum lift might have considerable effect on the manner in which a wing stalls for combinations with partial-span flaps.



(a) The $0.10c_w$ split flap.
(b) The $0.20c_w$ split flap.
(c) The $0.30c_w$ split flap.
(d) The $0.40c_w$ split flap.

FIGURE 11.—Comparison of profile-drag coefficients for airfoils with split flaps.

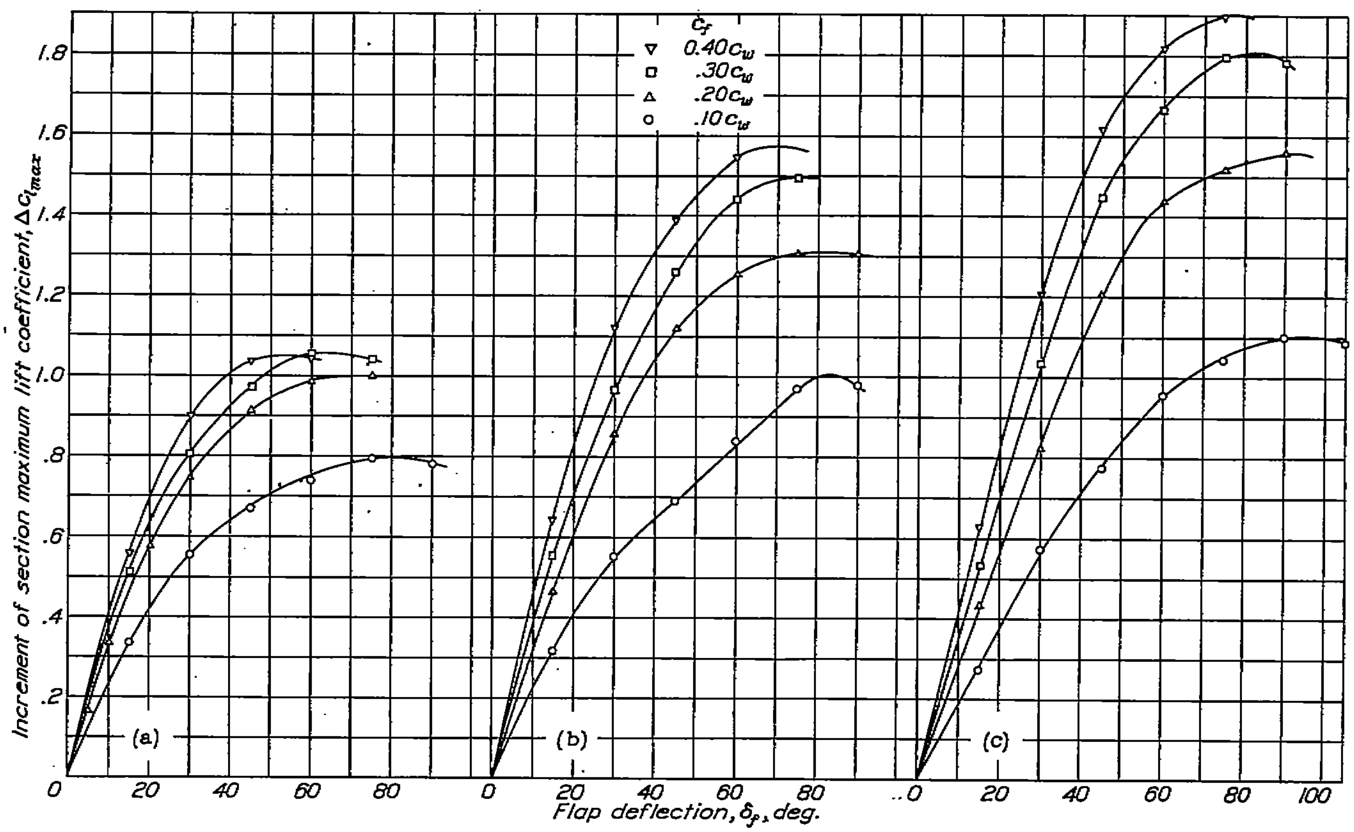
Similar section aerodynamic data are given for the N. A. C. A. 23021 airfoil with flaps in figure 9 and for the N. A. C. A. 23030 airfoil with flaps in figure 10. The angle of attack for maximum lift with the thicker airfoils with the flap deflected decreases with increasing thickness and flap chord to values as low as 5°, a change of about 10° from the plain airfoil. It should also be noted that a considerable increase in the profile-drag coefficient is obtained with increase in the flap chord.

The pitching-moment coefficient about the aerodynamic center increases quite rapidly with flap chord, flap deflection, and airfoil thickness. The marked increase with airfoil thickness is probably caused by the fact that the aerodynamic center is unusually far above the chord line and ahead of the quarter-chord point for the thick airfoils.

COMPARISON OF AIRFOILS WITH FLAPS

Effect on profile drag.—The effect of the $0.10c_w$ split flap on the profile drag of the three airfoils for various flap deflections is shown as envelope polar curves in figure 11 (a). Similar curves for the $0.20c_w$, the $0.30c_w$, and the $0.40c_w$ flaps are given, respectively, in figures 11 (b), 11 (c), and 11 (d). With the $0.10c_w$ flap, the drag is lowest throughout the complete lift range for the N. A. C. A. 23012 airfoil. The drag is lowest for the N. A. C. A. 23012 airfoil with the $0.20c_w$ flaps for lift coefficients less than 1.8; for lift coefficients greater than 1.8, it is lowest for the N. A. C. A. 23021 airfoil. The drag is lowest for the N. A. C. A. 23012 airfoil with the $0.30c_w$ and the $0.40c_w$ split flaps for lift coefficients less than about 2.1; for lift coefficients greater than 2.1, it is lowest for the N. A. C. A. 23021 airfoil. With the $0.30c_w$ and the $0.40c_w$ flaps, the drag is lower for the N. A. C. A. 23030 than for the N. A. C. A. 23012 airfoil for lift coefficients above 2.5.

A comparison of the parts of figure 11 shows the drag coefficients to be lowest for the smallest-chord flap suitable for a given lift coefficient for take-off.



(a) N. A. C. A. 23012 airfoil. (b) N. A. C. A. 23021 airfoil. (c) N. A. C. A. 23030 airfoil.

FIGURE 12.—Effect of split-flap deflection on increment of maximum lift coefficient for the various airfoils and flaps.

All the combinations with the split flap have higher drag coefficients throughout the take-off range than do the combinations with slotted flaps, which were developed for the N. A. C. A. 23012 and 23021 airfoils and are reported in references 1 and 3.

Effect on maximum lift.—The effect of deflecting the split flaps on the increment of section maximum lift coefficient $\Delta c_{l_{max}}$ is shown in figure 12, where $\Delta c_{l_{max}}$ is plotted against δ_f for all the combinations tested. The maximum $\Delta c_{l_{max}}$ increases with airfoil thickness for all the flap chords. The flap deflection for maximum $\Delta c_{l_{max}}$ decreases with increase in flap chord for any of the three airfoils. In figure 13, the maximum $\Delta c_{l_{max}}$ is plotted for each flap against flap chord for the three airfoils. The highest $\Delta c_{l_{max}}$ for the N. A. C. A. 23012 airfoil was obtained with the $0.30c_w$ flap, which was only slightly superior to the $0.20c_w$ flap on this airfoil. The highest $\Delta c_{l_{max}}$ for both the N. A. C. A. 23021 and the N. A. C. A. 23030 airfoils was obtained with the $0.40c_w$ flap. The $0.40c_w$ flap, however, gave little gain over the $0.30c_w$ flap, and probably no gain would be obtained by the use of a flap chord greater than $0.40c_w$ on the N. A. C. A. 23021 airfoil; for the N. A. C. A. 23030 airfoil, flaps of still larger chord might give a slight increase in $\Delta c_{l_{max}}$.

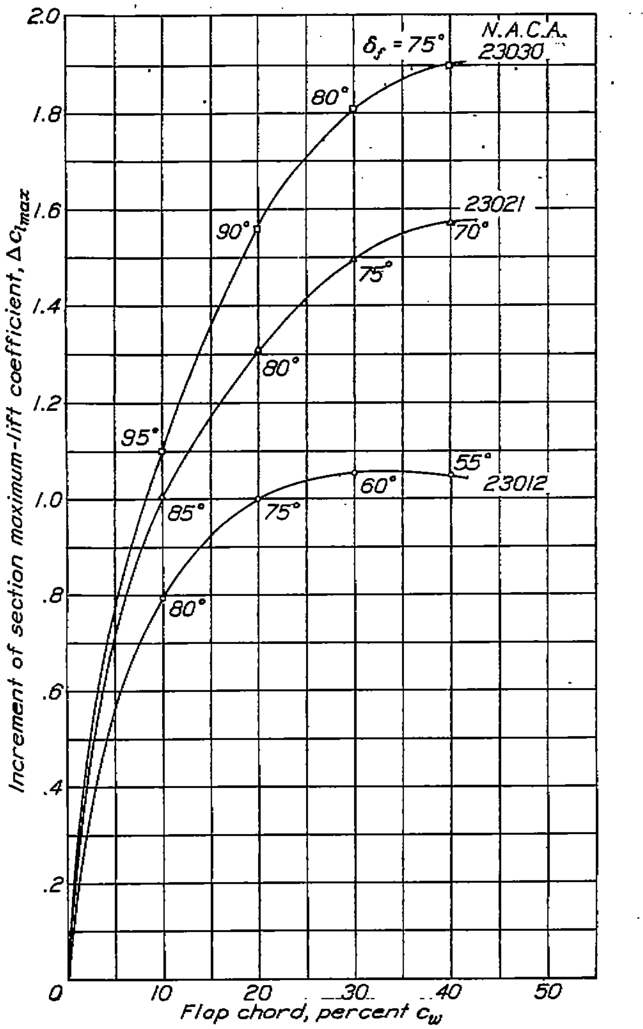


FIGURE 13.—Effect of chord of split flap on increment of maximum lift coefficient for three airfoil thicknesses.

The increments of maximum lift coefficient increase quite markedly with airfoil thickness; the values of $\Delta c_{l_{max}}$ vary from 1.05 for the N. A. C. A. 23012 to 1.9 for the N. A. C. A. 23030 airfoil. The final maximum lift coefficient, however, does not reflect this large difference in $\Delta c_{l_{max}}$, as is shown in figure 14, where $c_{l_{max}}$ for the plain airfoils and for the airfoils with flaps is plotted against airfoil thickness. The large loss in lift with thickness for the plain airfoil very nearly balances the large gain in increment of maximum lift with thickness for the airfoils with flaps. The final maximum lift coefficients for the N. A. C. A. 23012 and 23021 airfoils with the $0.10c_w$ flap was 2.34, which is about 8 percent higher than it was for the N. A. C. A. 23030 airfoil. The maximum lift coefficient for the airfoils with the $0.20c_w$ flap was 2.66 for the N. A. C. A. 23021 airfoil, which is about 4 percent higher than it was for the N. A. C. A. 23012 and 2 percent higher than it was for the N. A. C. A. 23030 airfoil. For the airfoils with the $0.30c_w$ and the $0.40c_w$ flaps, the maximum lift coefficient was 2.6 for the 23012 airfoil and increased about 11 percent with airfoil thickness for the 21-percent-thick airfoil. The maximum lift decreased slightly with thickness for the $0.30c_w$ flap and increased slightly for the $0.40c_w$ flap. The highest maximum lift coefficient, 2.94, was obtained with the $0.40c_w$ flap on the N. A. C. A. 23030 airfoil. In spite of the loss in lift of the plain airfoils with thickness, if for structural reasons wing thicknesses are increased to as much as 30 percent, no loss in ultimate section maximum lift coefficient will be encountered when split flaps with chords of $0.20c_w$ or larger are used.

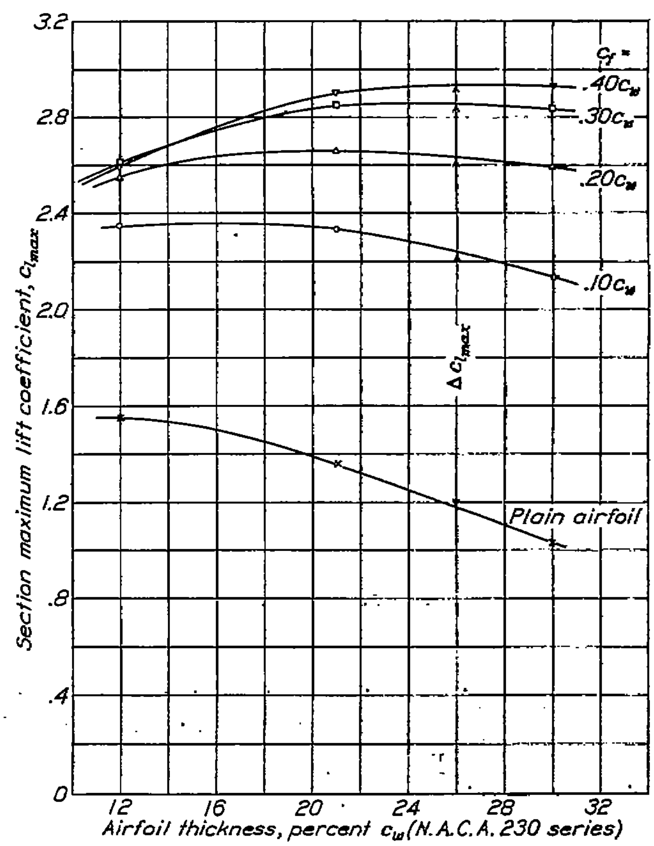


FIGURE 14.—Effect of airfoil thickness on maximum lift coefficient of N. A. C. A. 230 airfoils with and without split flaps.

SCALE EFFECT

The scale effect on maximum lift coefficients for the plain airfoils and the airfoils with flaps, over the range available in the 7- by 10-foot wind tunnel, is shown in figure 15, where $c_{l_{max}}$ is plotted against the value of R_e of the tests. This figure shows a very definite scale effect on the maximum lift coefficient for the N. A. C. A. 23012 airfoil with or without flaps but shows practically none for the N. A. C. A. 23021 and 23030 airfoils with or without flaps. The increment of maximum lift coefficient is therefore practically independent of scale over the range that could be investigated.

APPLICATION OF OTHER AIRFOILS

The maximum lift coefficients for airfoils of the N. A. C. A. 430 and 630 series with split flaps may be computed with satisfactory accuracy by adding the

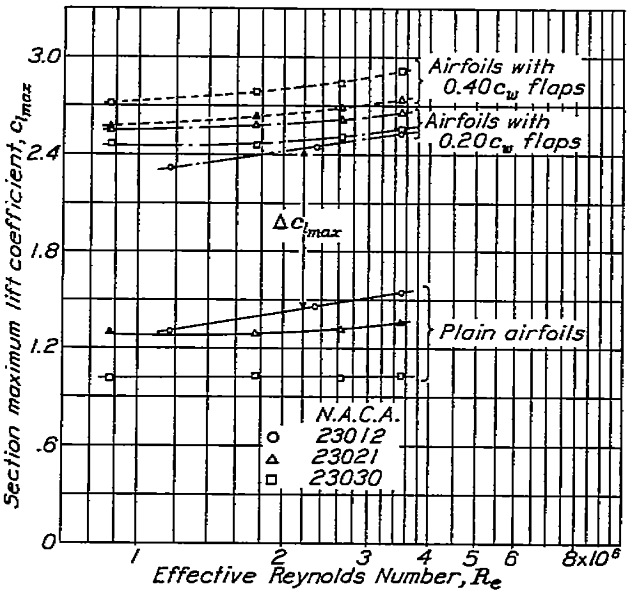


FIGURE 15.—Effect of scale on maximum lift coefficient of three airfoils with and without split flaps; 7- by 10-foot wind tunnel.

$\Delta c_{l_{max}}$ for the proper flap chord and airfoil thickness from the 230 series to the $c_{l_{max}}$ of the plain airfoil under consideration. This procedure is justified for thicknesses from 9 to 21 percent, as indicated in reference 4. The same procedure would also probably be satisfactory for other airfoils with the position of maximum camber near the leading edge. It should be remembered in applying these data that they are section characteristics and that these maximum lift coefficients cannot be realized on a wing of finite span unless it is designed so that all sections reach maximum lift simultaneously.

CONCLUDING REMARKS

Aerodynamic data are made available for airfoils 12 to 30 percent thick with split flaps having chords 10 to 40 percent of the wing chord. The final maximum lift coefficients for the three airfoils tested with the $0.20c_w$ flap were about equal; for the airfoils with the $0.10c_w$ flap, the maximum lift coefficient decreased with airfoil thickness; and for the airfoils with the $0.30c_w$ and the $0.40c_w$ flaps, the maximum lift coefficient increased with airfoil thickness.

Within the range covered, the increment of maximum lift coefficient due to the split flaps was practically independent of scale. The profile-drag coefficient increased quite rapidly with thickness for the plain airfoils and was about twice as large for the N. A. C. A. 23030 as for the 23012 airfoil.

LANGLEY MEMORIAL AERONAUTICAL LABORATORY,
NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS,
LANGLEY FIELD, VA., *March 10, 1939.*

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TABLE I

ORDINATES FOR N. A. C. A. 230 AIRFOILS

[Stations and ordinates in percent of wing chord]

Station	23012		23021		23030	
	Upper surface	Lower surface	Upper surface	Lower surface	Upper surface	Lower surface
0	-----	0	-----	0	4.82	0
1.25	2.67	−1.23	4.87	−2.08	7.37	−2.63
2.5	3.61	−1.71	6.14	−3.14	8.90	−4.27
5	4.91	−2.26	7.93	−4.52	11.06	−6.54
7.5	5.80	−2.61	9.13	−5.55	12.57	−8.28
10	6.43	−2.92	10.03	−6.32	13.68	−9.65
15	7.19	−3.50	11.19	−7.51	15.20	−11.52
20	7.50	−3.97	11.80	−8.30	16.07	−12.61
25	7.60	−4.28	12.05	−8.76	16.46	−13.20
30	7.55	−4.46	12.06	−8.95	16.57	−13.46
40	7.14	−4.48	11.49	−8.83	15.89	−13.13
50	6.41	−4.17	10.40	−8.14	14.38	−12.11
60	5.47	−3.67	8.90	−7.07	12.34	−10.47
70	4.36	−3.00	7.09	−5.72	9.86	−8.42
80	3.08	−2.16	5.05	−4.13	7.03	−6.09
90	1.68	−1.23	2.76	−2.30	3.87	−3.40
95	.92	−.70	1.53	−1.30	2.15	−1.86
100	.13	−.13	.22	−.22	.32	−.32
L. E. radius	1.58		4.85		9.90	

Slope of radius through end of chord: 0.305